Amendments to the Specification:

Please replace paragraph [0004] with the following amended paragraph:

[0004] The high-thrust propulsion system can comprise a liquid apogee engine (LAE) that burns a mixture of hydrazine and oxidizer to generate a thrust of about 100 to 150 lbf with a mass specific impulse (ISP) of about 325 seconds (the ISP is the ratio of the thrust to the mass flow rate, a higher value indicating that less fuel is needed to achieve the same applied impulse). The orbit transfer then can be performed ~~by firing a sequence~~ <u>by performing a sequence</u> of four to six LAE maneuvers at or near the orbit apogee, with each maneuver lasting less than 1 hour. Using this approach, the transfer to the GEO mission orbit can be accomplished in less than 2 weeks. However, the performance of this approach, as measured by the mass delivered to orbit on a given launch vehicle, is limited by the capability of the launch vehicle and the amount of chemical propellant that can be stored on-board the spacecraft.

Please replace paragraph [0005] with the following amended paragraph:

[0005] The LAE orbit transfer approach provides reasonable efficiency and a rapid orbit transfer, but for heavy spacecraft the combined launch vehicle and on-board LAE system can be insufficient to transfer the spacecraft to the mission GEO orbit. To address this problem, a hybrid orbit transfer approach recently has been introduced that performs the orbit transfer ~~is~~ in two phases. The first phase includes some number of LAE firings to transfer the spacecraft from the initial injection orbit to an intermediate orbit, which typically has its perigee above the Van Allen radiation belts to limit solar array degradation during the final orbit transfer phase (phase 2).

Please replace paragraph [0012] with the following amended paragraph:

[0012] Thus, an improved method is desired that may be used to determine thrust trajectory and on/off firing regions in a way that provides ~~the~~ <u>a</u> ~~improved~~ fuel use reduction with minimum impact to the total orbit transfer time. Preferably, the method does not require the use of assumptions regarding the functional form of the thrust trajectory or the orbital locations where

firing should occur. Also, a system is desired for implementing such a trajectory and firing regions on-board a spacecraft to perform a transfer from an intermediate orbit to a final mission orbit.

Please replace paragraph [0030] with the following amended paragraph:

[0030] The present invention relates to ~~a~~ orbit transfer techniques, and more particularly to systems and methods for efficiently transferring a spacecraft from one orbit to another orbit.

Please replace paragraph [0034] with the following amended paragraph:

[0034] The location and position of thrusters 116 and 118 on spacecraft 100 are for illustration purposes only. One skilled in the art will appreciate that the thrusters can be located at a number of different locations on the spacecraft, which can be dictated by the spacecraft mission and orbit transfer requirements. Also, the spacecraft illustrated in Fig. 1 is but one example of a spacecraft that may embody the systems and methods of the present invention. Therefore, the present invention is not limited to the illustrated embodiment. One skilled in the art will appreciate that the systems and methods of the present invention can be used on any spacecraft configuration and for any spacecraft mission.

Please replace paragraph [0036] with the following amended paragraph:

[0036] The present invention provides for systems and methods for improving the efficiency of transferring a spacecraft from an initial orbit to a final orbit. In accordance with the present invention, an optimal orbit transfer trajectory is solved for starting from the prior-art continuous-firing solution. Next, the system determines when thrusters can and/or should be turned-off by calculating when Gamma is less than a specified threshold value. The threshold value is determined based on a number of factors, including desired fuel savings and maximum orbit transfer time available. A numerical iteration is performed to determine a minimum-time orbit transfer trajectory that achieves the final orbit with the Gamma constraint imposed.

Please replace paragraph [0037] with the following amended paragraph:

[0037] Based on the results of the numerical optimization calculations, the spacecraft attitude is controlled to orient the thrusters' thrust vector in the direction of the specified thrust trajectory vector. The thrusters then are modulated so they fire within the specified firing regions of each orbit and do not fire in the specified off regions. To carry out an orbit transfer, the present invention can determine parameters for thruster firings to achieve the desired orbital change most efficiently. The thruster parameters can include when to turn thrusters on and off, and other parameters which are either the thrust vector direction (in an inertial or any other coordinate frame) or some other parameters which enable the spacecraft on-board software to compute the thrust vector direction at any time. A more detailed discussion of the systems and methods of the present invention is given below.

Please replace paragraph [0041] with the following amended paragraph:

[0041] where λ is a 6x1 vector $\left(\lambda^T = [\lambda_1, \lambda_2, \lambda_3, \lambda_4, \lambda_5, \lambda_6]\right)$, which is the co-state vector of the minimum-time orbit transfer solution. The co-state vector is a function of time and varies at each time step over the orbit transfer trajectory. The vector λ is a Lagrange multiplier that multiplies ~~with~~ the constraint equations in the formulation of the minimum-time orbit transfer problem. As is known to those of skill in the art, the constraint equations are the equations of motion of the orbit elements, which are also referred to as the orbit perturbation equations.

Please replace paragraph [0042] with the following amended paragraph:

[0042] The orbit element state vector z is a 6x1 vector that contains ~~equations for:~~ (1) the orbit semi-major axis; (2) the y-component of eccentricity vector; (3) the x-component of eccentricity vector; (4) the y-component of inclination vector; (5) the x-component of inclination vector; and (6) the eccentric longitude (x and y are with respect to the equinoctial coordinate frame). The orbit state vector is as follows:

Please replace paragraph [0044] with the following amended paragraph:

[0044] where the orbit element perturbation equations for a, h, k, p, q, F are well known in the art. Each element of λ multiplies ~~with~~ a corresponding orbit element perturbation equation. In particular, λ_6 is the 6th element of λ and multiplies the perturbation equation for the eccentric longitude F, which is the 6th element of the orbit element state vector z. In Equation (1), $\dot{z}$ is the 6x1 vector of orbit element time derivatives. Also, the quantity $\dot{F}$ in Equation (1) denotes the time rate of change of the eccentric longitude F and is also the 6th element of $\dot{z}$.

Please replace paragraph [0052] with the following amended paragraph:

[0052] The minimum-time orbit transfer trajectory vector can be calculated using an iterative operation to determine the initial co-state vector λ, and the minimum transfer time from the initial orbit to the final orbit, while imposing the thruster on/off constraint of $\Gamma \leq \Gamma_0$. In accordance with one embodiment of the invention, the initial co-state vector λ and the minimum transfer time can be obtained using a Newton-Raphson iteration approach, in which ~~the initial values~~ the iteration starting values for the co-state vector and the minimum transfer time can be the values which form the solution for the continuous-fire orbit transfer problem.